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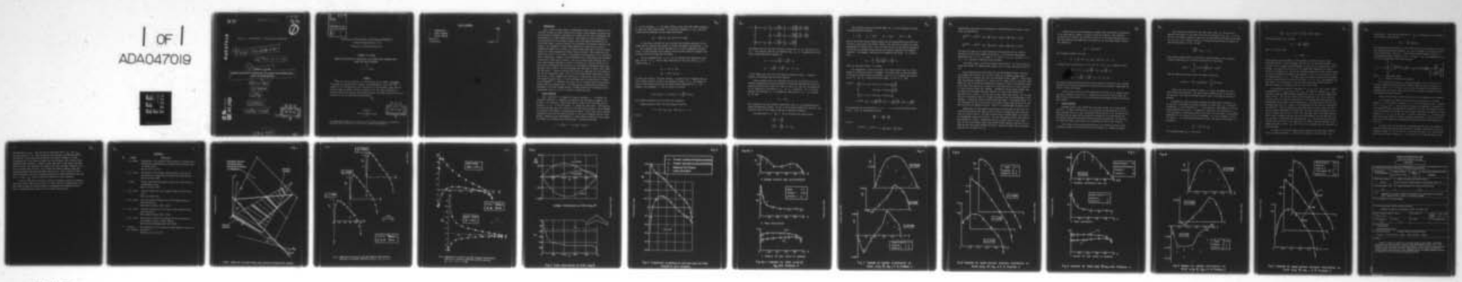

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EUROMECH 75 LECTURE:

ITERATIVE CALCULATION OF SUBCRITICAL FLOW AROUND THICK CAMBERED WINGS: DIRECT AND DESIGN PROBLEMS

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(10) C. C. L. Sells

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EUROMECH 75 LECTURE:

ITERATIVE CALCULATION OF SUBCRITICAL FLOW AROUND THICK CAMBERED WINGS: DIRECT AND DESIGN PROBLEMS

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C. C. L. Sells

SUMMARY

This is a written version of a lecture given by the author at Euromech Colloquium 75 which was held at Rhode, near Braunschweig in May 1976. The subject of the colloquium was 'The calculation of flow fields by panel methods' and this paper describes iterative calculations of the classic problems of steady inviscid flow around a thick cambered wing, providing improved accuracy in relation to the so-called RAE Standard Method.

This Memorandum expresses the opinions of the author and does not necessarily represent the official view of the Royal Aircraft Establishment.

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1 INTRODUCTION

Over a period of many years, an approximate method for the solution of the classical problem of steady inviscid flow around a thick cambered isolated wing was developed in England, principally by Dr Weber and the late Dr Küchemann, and became known as the RAE Standard Method. This method is based effectively on the representation of the wing by planar source and doublet distributions; in first-order theory, sources represent wing thickness and doublets represent wing loading. The strengths of these planar singularity distributions could be estimated by appeal to the theory of infinite swept wings and of infinite kinked wings, with some interpolation between relevant regions. This method was relatively fast, and as it was based on the physics of the problem, it was relatively easy to understand. However, in time questions were raised whether the estimates involved were sufficiently accurate. One way to answer these questions was, to write some computer programs to evaluate accurately the velocity fields due to given planar singularity distributions, by performing the necessary double integrations, and in 1968 to 1970 such programs were developed at RAE by John Ledger[1] and the author[2]. Separate versions were written to calculate the flow fields on and off the singularity plane. Next, in 1972 Johanna Weber[3] showed how to apply the results of these computations iteratively to improve the basic solution to at least second-order accuracy. In the following years the author wrote a further set of programs[4,5] to implement her ideas, and it is these iterative calculation methods that are described here. The whole group of methods thus represents a refinement of the RAE Standard Method, although because of the numerical double integrations there is a price to be paid in computing time.

2 DIRECT PROBLEMS

Fig 1 shows half of a symmetrical wing at incidence α . We define the cartesian coordinate y measured to starboard from the plane of symmetry, and at each y we consider a plane containing the local section chordline and at incidence $\alpha + \alpha_T$ to the free stream, α_T being the local twist. The singularity distributions, which are actually in the wing chordal surface, can be thought of as lying in this plane, to second-order accuracy. We can take local cartesian coordinates (x,z) with the x-axis parallel to the local chordline and the z-axis upwards, completing the right-handed set. Wing surface ordinates in this system now vanish at the leading and trailing edges. We define them thus:

$$z = z_w(x,y) = \pm z_t(x,y) + z_s(x,y) \quad .$$

z_t is the thickness, z_s the camber ordinate; upper and lower signs correspond to upper and lower wing surfaces. For the local incidence $\alpha + \alpha_T$, the free stream flow is normalized to unit speed and written:

$$\underline{U}_\infty = [\cos(\alpha + \alpha_T),\ 0,\ \sin(\alpha + \alpha_T)] \ .$$

In the direct problem[4], we aim to satisfy the boundary condition of zero normal flow by iterative calculation of source and doublet distributions q and ℓ. Let us suppose that we have obtained a set of trial singularity distributions. We now calculate their separate velocity fields using the computer subroutines written by Ledger and the author.

For an uncambered wing, $z_s = 0$, we can calculate the velocities on the upper surface $z = z_t$ and use these relations for the source field $\underline{u}_t$ and doublet field $\underline{u}_\ell$

$$\underline{u}_t = (u_t,\ v_t,\ \pm w_t)$$

$$\underline{u}_\ell = (\pm u_\ell,\ \pm v_\ell,\ w_\ell) \ ,$$

to obtain the values on the lower surface. In general, for a cambered wing, we still compute velocity fields on the thickness surface $z = z_t$ and obtain the values on the actual wing surface by Taylor series expansions in z_s : for example, the streamwash u_t due to sources is written:

$$u_t(x,y,z_t \pm z_s) \doteqdot u_t(x,y,z_t) \pm z_s \frac{\partial u_t}{\partial z}(x,y,z_t)$$

with similar expressions for the other five components.

Substituting all these into the boundary condition

$$R \equiv (\underline{U}_\infty + \underline{u}_t + \underline{u}_\ell) \ . \ \mathrm{grad}\ (z_w - z) = 0$$

we get

$$R \equiv \left[\cos(\alpha + \alpha_T) + \left(u_t \pm z_s \frac{\partial u_t}{\partial z}\right) \pm \left(u_\ell \pm z_s \frac{\partial u_\ell}{\partial z}\right)\right]\left(\pm \frac{\partial z_t}{\partial x} + \frac{\partial z_s}{\partial x}\right)$$
$$+ \left[\left(v_t \pm z_s \frac{\partial v_t}{\partial z}\right) \pm \left(v_\ell \pm z_s \frac{\partial v_\ell}{\partial z}\right)\right]\left(\pm \frac{\partial z_t}{\partial y} + \frac{\partial z_s}{\partial y}\right)$$
$$- \left[\sin(\alpha + \alpha_T) \pm \left(w_t \pm z_s \frac{\partial w_t}{\partial z}\right) + \left(w_\ell \pm z_s \frac{\partial w_\ell}{\partial z}\right)\right] = 0 \ .$$

We multiply this out and take the plus or minus parts to R_t and the rest to R_ℓ. In the special case of an uncambered wing, $z_s = 0$, assuming small incidence, these expressions simplify to

$$R_t = (1 + u_t) \frac{\partial z_t}{\partial x} + v_t \frac{\partial z_t}{\partial y} - w_t$$

$$R_\ell = u_\ell \frac{\partial z_t}{\partial x} + v_\ell \frac{\partial z_t}{\partial y} - w_\ell - (\alpha + \alpha_T) \ .$$

In this simple case, the source and doublet problems uncouple. In general, however, the problems must be solved together.

These residual errors are evaluated, and tell us not only how well the boundary conditions have been satisfied, but also what perturbation source and doublet distributions to add on in order to cancel R_t and R_ℓ approximately. R_t is regarded as a shortfall in w_t (if R_t is positive, w_t is not big enough), and similarly R_ℓ is regarded as a shortfall in w_ℓ. So to cancel R_t we take

$$\Delta q = 2R_t \ ,$$

and we generate the extra doublet distribution from R_ℓ by an approximate but rapid lifting-surface theory. We use a vortex-lattice method for that job. The iteration cycle is now complete, and can be repeated as desired.

Starting values of q and ℓ can be obtained from linear theory:

$$R_t^{(0)} = \frac{\partial z_t}{\partial x}$$

$$R_\ell^{(0)} = \frac{\partial z_s}{\partial x} - (\alpha + \alpha_T) \ .$$

For subcritical flow at low Mach number M_∞, we can transform to affine or Prandtl-Glauert variables:

$$x = \tilde{x}\beta \qquad u_t = \tilde{u}_t/\beta \qquad u_\ell = \tilde{u}_\ell/\beta \qquad (\beta^2 = 1 - M_\infty^2)$$

Then in the affine $(\tilde{x},y,z)$ space we again have an incompressible flow which can be built up from source and doublet fields. Note that this accounts for linear compressibility effects only. Note also that the flow is not exactly equivalent to flow over an analogous wing, because the boundary conditions are slightly different. For example, considering the uncambered wing again, R_t is given thus:

$$R_t = \left(1 + \frac{\tilde{u}_t}{\beta}\right) \frac{1}{\beta} \frac{\partial z_t}{\partial \tilde{x}} + v_t \frac{\partial z_t}{\partial y} - w_t \quad .$$

There are two extra factors β present.

The computation of velocity fields on the wing surface, even at a finite number of collocation points, is lengthy. So we would like to keep the number of iteration cycles as small as possible. One trick is to expand the residual errors R_t and R_ℓ as Maclaurin series (about $z = 0$) in z_w thus:

$$\begin{aligned} R^{(n,p+1)} &= \left[\cos(\alpha + \alpha_T) + u(x,y,z_w) + \Delta u(x,y,z_w)\right] \frac{\partial z_w}{\partial x} \\ &\quad + \left[v(x,y,z_w) + \Delta v(x,y,z_w)\right] \frac{\partial z_w}{\partial y} \\ &\quad - \left[\sin(\alpha + \alpha_T) + w(x,y,z_w) + \Delta w(x,y,z_w)\right] \\ &= R^{(n,p)} + \left(\Delta u + z_w \frac{\partial \Delta u}{\partial z}\right) \frac{\partial z_w}{\partial x} + \left(\Delta v + z_w \frac{\partial \Delta v}{\partial z}\right) \frac{\partial z_w}{\partial y} - \left(\Delta w + z_w \frac{\partial \Delta w}{\partial z}\right) \quad . \end{aligned}$$

All quantities are now evaluated on $z = 0$. p is an inner iteration superscript. Since (in incompressible flow)

$$\frac{\partial \Delta w}{\partial z} = - \frac{\partial \Delta u}{\partial x} - \frac{\partial \Delta v}{\partial y}$$

we have

$$R^{(n,p+1)} = R^{(n,p)} - \Delta w + \frac{\partial}{\partial x}(z_w \Delta u) + \frac{\partial}{\partial y}(z_w \Delta v) \quad .$$

Substituting everything in, multiplying out and separating the ± parts, we get these two expressions:

$$R_t^{(n,p+1)} = R_t^{(n,p)} - \Delta w_t + \frac{\partial}{\partial x}(z_t \Delta u_t + z_s \Delta u_\ell) + \frac{\partial}{\partial y}(z_t \Delta v_t + z_s \Delta v_\ell)$$

$$R_\ell^{(n,p+1)} = R_\ell^{(n,p)} - \Delta w_\ell + \frac{\partial}{\partial x}(z_t \Delta u_\ell + z_s \Delta u_t) + \frac{\partial}{\partial y}(z_t \Delta v_\ell + z_s \Delta v_t) \quad .$$

The first two terms on the right sides have been cancelled by the iterative selection of source and doublet distributions. We can now use approximate expressions for Δu_t, Δv_t in terms of Δq and for Δu_ℓ, Δv_ℓ in terms of $\Delta \ell$, on $z = 0$, instead of computing them accurately.

For simple wings, two main iterations often suffice. For curved wings, or wings with cranks, or complex camber shapes, three or more iterations should be run.

Some comparisons have been made with the BAC Neumann program of Roberts, the standard datum method, for the first family of uncambered wings taken by BAC and NLR as a standard test case, and discussed by Hunt (BAC)[7,8]. These wings have the planform of RAE Wing 'A', aspect ratio 6, taper ratio $\frac{1}{3}$, mid-chord sweep angle 30^o, and the chordwise thickness distribution is that of the NACA 00 series. Fig 2 shows some results for a wing with a 15% thick section, at zero incidence, at three stations across the semispan. Agreement is good, except perhaps near the thick trailing-edge; Hunt has, however, now told us that Roberts had extended this to a point. The top part of Fig 3 shows some results, for the same wing at 5^o incidence; there is a slight discrepancy near the leading-edge in mid-semispan. Perhaps the author's method needs more than 11 collocation points. On the bottom part of Fig 3 are results for the much thinner wing, 2% thick, at the same incidence and spanwise station. There is no leading-edge discrepancy here.

Comparisons with the Roberts method have also been made for RAE Wing 'B', of which the planform (the same as that of Wing 'A'), camber and twist distributions are shown in Fig 4. For zero incidence and zero Mach number, the results in mid-semispan compare well (Fig 5). We also show results from a fast approximate method due to Lock. Comparisons are not so good[4] near the root, where Wing 'B' has local dihedral which the present method neglects, and near the tip where neither the present method nor Roberts' method satisfies the end boundary condition on the square cut tip.

Ground effect can be represented by image source and doublet distributions, of the same sign for images, of opposite sign for doublets. The iteration proceeds much as before; the velocity fields due to the image distributions are also evaluated by the Ledger-Sells subroutine on the real wing chordal surface, and combined as

$$\underline{u}_G = (u_G, v_G, w_G) \quad .$$

The boundary condition now reads

$$R' \equiv (\underline{U}_\infty + \underline{u}_t + \underline{u}_\ell + \underline{u}_G) \cdot \text{grad}\,(z_w - z) = 0 \quad .$$

As before, this can be split as $\pm R_t + R_\ell = 0$. For $z_s = 0$ again, we have

$$R'_t = (1 + u_t + u_G)\frac{\partial z_t}{\partial x} + (v_t + v_G)\frac{\partial z_t}{\partial y} - w_t$$

$$R'_\ell = u_\ell \frac{\partial z_t}{\partial x} + v_\ell \frac{\partial z_t}{\partial y} - w_\ell - w_G - (\alpha + \alpha_T) \quad .$$

For simple wings the program now needs three iterations instead of two, probably because the perturbation singularity distributions are generated, neglecting the effect of their own images.

In the current version, we have neglected variations in $\underline{u}_G$ between upper and lower surfaces for each (x,y). Comparison with results from the Hess and Smith surface panel method for a two-dimensional wing suggests that this may have been unwise[6].

3 DESIGN PROBLEMS

Programs based on this method have also been written for various design options[5]. In order of description, we can specify the thickness and the first-order loading (that is to say, the doublet) distributions; or we can specify the thickness and the upper-surface pressure distribution; or we can specify the first-order loading and the upper-surface pressure distribution. In each case, the camber and twist distributions are determined as part of the solution.

In all cases, the source distribution has to be found, and this is done iteratively using the successive error field iterates R_t exactly as before.

The successive iterates for the other error field R_ℓ are now used to generate corrections to the camber and twist distributions Δz_s, $\Delta\alpha_T$. If we go back to the boundary condition, substitute the new camber and twist distributions $z_s + \Delta z_s$, $\alpha_T + \Delta\alpha_T$, neglect products of perturbation quantities and equate the resulting expression to zero, we get:

$$\frac{\partial \Delta z_s}{\partial x} - \Delta\alpha_T = -R_\ell \quad .$$

Since wing ordinates vanish at leading and trailing edges in our coordinate system, integrating over the whole chord gives the wing twist:

$$\Delta\alpha_T(y) = \int_{x_L}^{x_T} R_\ell(x',y)dx' \Big/ \left[x_T(y) - x_L(y)\right] \quad .$$

Then the indefinite integral gives the camber correction:

$$\Delta z_s(x,y) = \left[x - x_L(y)\right]\Delta\alpha_T(y) - \int_{x_L}^{x} R_\ell(x',y)dx' \quad .$$

This is all that is needed for Option 1, in which thickness and first-order loading are specified. The upper-surface pressures come out as part of the solution. This option converges quickly, about as quickly as the direct program.

For Option 2, when we specify thickness and upper-surface pressure C_{pu}, we now have to calculate the doublet strength to satisfy the C_{pu} condition iteratively. A first estimate for ℓ is provided by a modification of the RAE Standard Method due to Lock. In subsequent iterations, the shortfall in C_{pu} is in principle a second-order quantity and from it we can derive a second-order expression for the correction to the streamwash Δu_ℓ due to loading. We will use suffix u to represent upper-surface values. The current local speed Q_u is given by

$$Q_u^2 = U_u^2 + V_u^2 + W_u^2 \quad .$$

For the design speed $\bar{Q}_u$ we can write:

$$\bar{Q}_u^2 = (U_u + \Delta u_\ell)^2 + V_u^2 + W_u^2 \cong Q_u^2 + 2U_u\Delta u_\ell \quad .$$

From this equation Δu_ℓ follows:

$$\Delta u_\ell = (\bar{Q}_u^2 - Q_u^2)/2U_u \quad .$$

Then $\Delta\ell$ is given thus:

$$\Delta\ell = 4\beta\Delta u_\ell \quad .$$

This iteration cycle converges in mid-semispan, and with the help of some relaxation factors it also converges near the tip, for the cases studied. But near the root of a swept wing convergence is slow, and the iteration may in fact diverge. It seems that near the root the logarithmic singularity of linear theory in the upwash Δw_ℓ affects the velocity components so that a small change $\Delta\ell$ does not necessarily produce small changes in v_ℓ, w_ℓ and so does not have the desired effect on Q_u. For some cases, we avoided instability by arranging an inner iteration cycle in which we calculate Δv_ℓ and Δw_ℓ approximately and then update Q_u and generate a further correction doublet field as before. This also speeds up convergence over the rest of the wing.

The program was applied to RAE Wing 'B' at $M_\infty = 0.8$, as a check. Target distributions of ℓ, C_{pu}, section lift C_{LL} and centre of pressure x_{cp} were generated using the earlier design program (for Option 1), and are shown as full lines in Fig 6. The top part shows the convergence near the root of the load function (this is a function made regular at the leading and trailing edges by multiplying out the square root singularities). The load function has not quite converged near the root. However, in mid-semispan and near the tip convergence is excellent[5]. This is reflected in the lower parts of Fig 6, where the twist and the section aerodynamic quantities are very well converged outboard, but not quite on target near the root. Fig 7 tells the same story: the camber distribution is well converged near the tip and in mid-semispan, but has some way to go near the root. Fig 8 shows the same again for C_{pu}, but we see that the final iterates are very close to the target near the root, in marked contrast to the camber and twist, so that C_{pu} near the root is evidently not very sensitive to the wing shape.

For Option 3, we specify upper-surface pressure and first-order loading. Here, it is the thickness z_t that must be adjusted to get the right C_{pu}

distribution. This time the shortfall in C_{pu} is converted into a shortfall in velocity due to sources thus:

$$\Delta u_t = (\bar{Q}_u^2 - Q_u^2)/2U_u \ .$$

The right side of this equation is the same as that for the doublets in Option 2. The required perturbation thickness Δz_t can be thought of as giving an extra source distribution $2\partial\Delta z_t/\partial x$. From the RAE Standard Method we have the following approximate formula for Δu_t in terms of Δz_t (the simplified relation for incompressible flow is given here):

$$\Delta u_t = \cos\Lambda \left[\frac{1}{\pi} \int_{x_L}^{x_T} \frac{\partial\Delta z_t(x',y)}{\partial x'} \frac{dx'}{x - x'} - K_2(y)\, \frac{1}{\pi} \ln \frac{1 + \sin\Lambda}{1 - \sin\Lambda} \, \frac{\dfrac{\partial\Delta z_t}{\partial x}}{\left[1 + \left(\dfrac{\partial\Delta z_t}{\partial x}\right)^2\right]^{\frac{1}{2}}} \right]$$

with Λ = local sweep angle

K_2 = spanwise interpolation parameter.

In a sheared-wing region, or near mid-semispan, $K_2 = 0$ and this is an ordinary Cauchy-type principal-value integral equation, and the solution is standard. When K_2 is not zero, we can solve this equation iteratively, by regarding the last term as known from a previous iterate, and again inverting the standard integral equation.

This option converges slowly near the root, but reasonably well elsewhere.

It is possible to combine the second and third options into a hybrid Option 4. In this option, a station $y = y^*$ is picked near the root. Outboard of $y = y^*$, C_{pu} and z_t are specified, leading to a sequence of iterates for the outboard doublet distribution as in Option 2, and then the doublet distribution is constrained to exhibit the same chordwise behaviour throughout the region inboard of $y = y^*$, up to the root. To obtain a specified C_{pu} in this region also, we can now adjust the inboard thickness distribution z_t as in Option 3. This option thus avoids the convergence problems associated with doublets near the root.

As a test case for this option, a modified wing '$\hat{B}$' was first designed to have a suitable load distribution and RAE 101 thickness distribution, 9% thick outboard, rising to 13.5% at the root. This gave an output C_{pu} distribution which was then input to Option 4. Fig 9 shows as full lines the known target

distributions of z_t/c near the root, of wing twist and of C_{LL} and x_{cp}. The first guess at z_t/c was the same as that outboard, the 9% thick RAE 101 distribution. After four more iterations the inboard thickness is slowly converging, but is not quite on target yet, and we expect this to show up when comparing other quantities. Indeed, in the lower part of the figure the twist and the section lift and centre of pressure are well converged outboard, but not quite on target near the root. Fig 10 shows the camber iterates: again well converged outboard, but not quite home near the root. Fig 11 shows the upper-surface pressures: also well converged outboard, and slight differences near the root. In fact, C_{pu} does not seem very sensitive to root thickness.

It can be seen that this Option 4 gives the designer a chance to maintain good flow quality right into the root of a swept wing, by increasing the thickness which he wants to increase anyway to strengthen the wing-body junction. Thus, it seems that this could be a quite useful design option.

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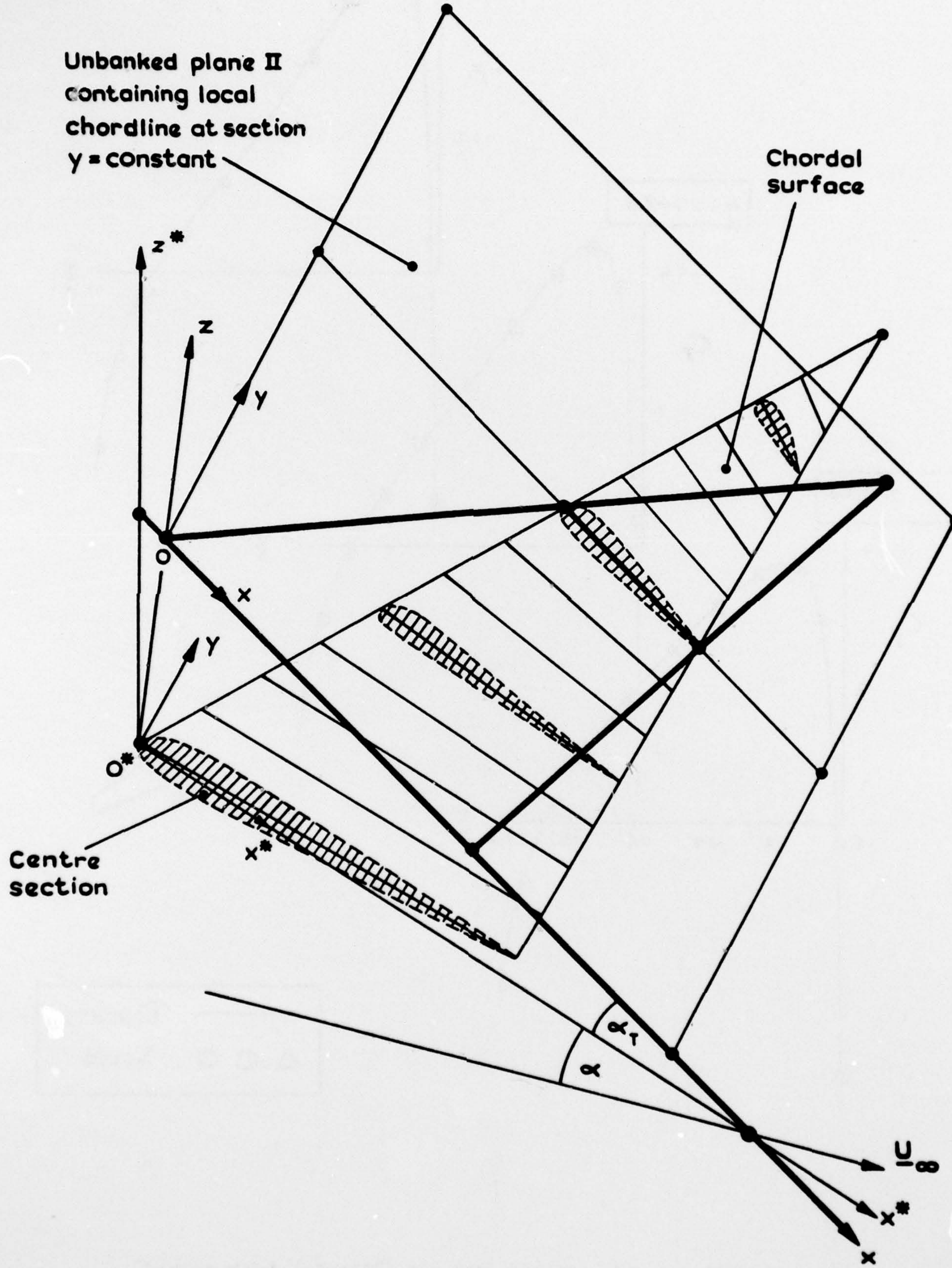


Fig.1 Sketch of half wing and typical singularity plane

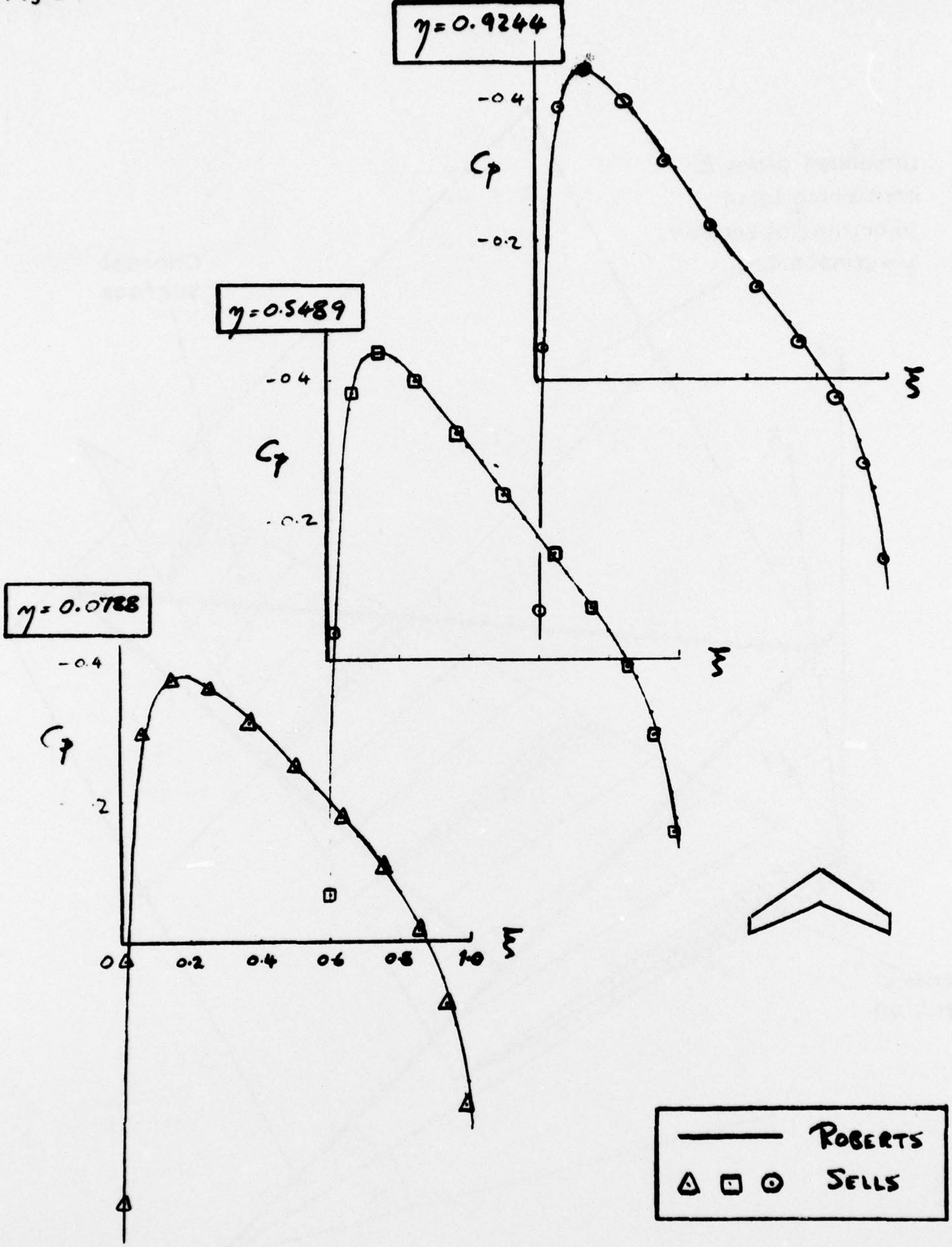


Fig 2 Comparison with results from BAC (Roberts) datum method.
RAE Wing 'A', planform, NACA 0015 section, $M_\infty = 0$, $\alpha = 0$

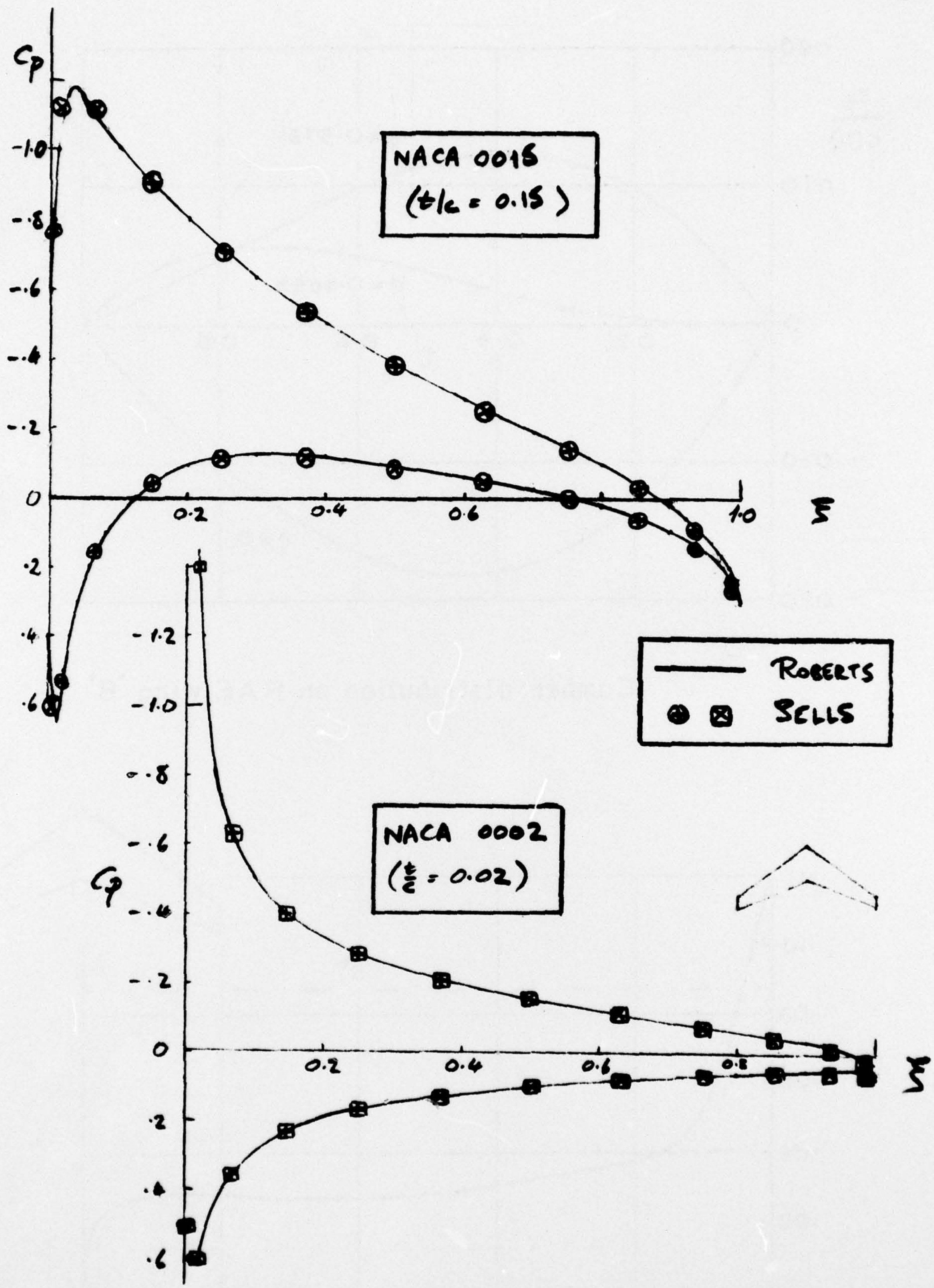
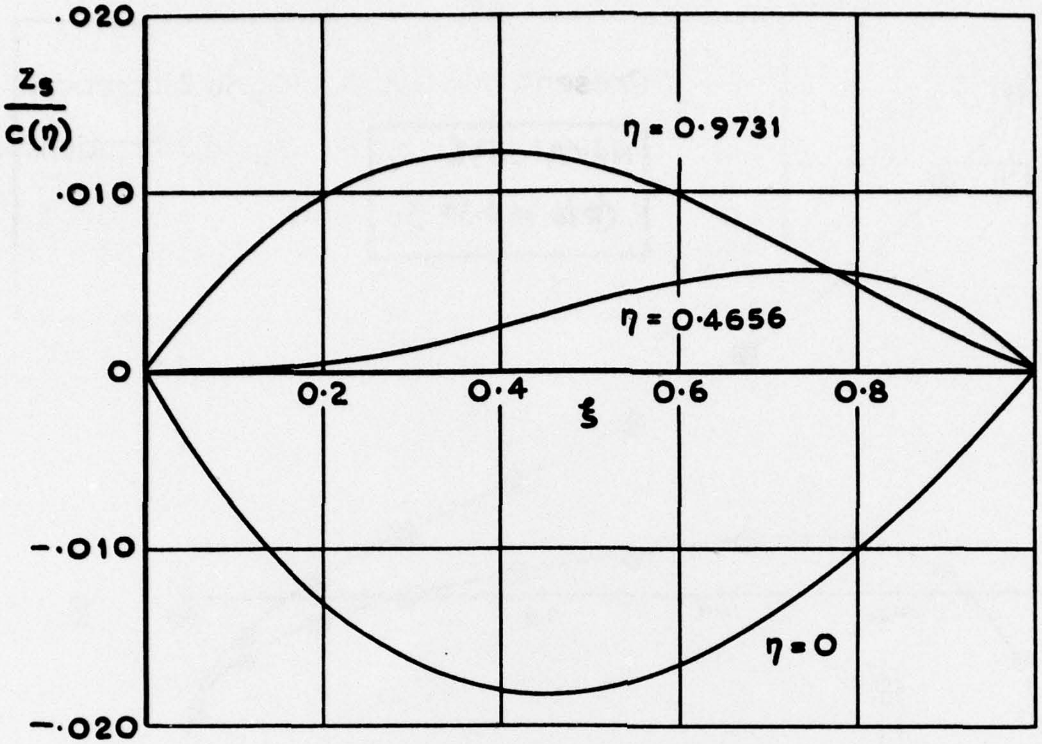


Fig 3 Comparison with results from BAC (Roberts) datum method. RAE Wing 'A' planform, NACA 0002 and 0015 sections, $M_\infty = 0$, $\alpha = 5^o$, $\eta = 0.549$

Fig. 4



Camber distribution on RAE wing 'B'

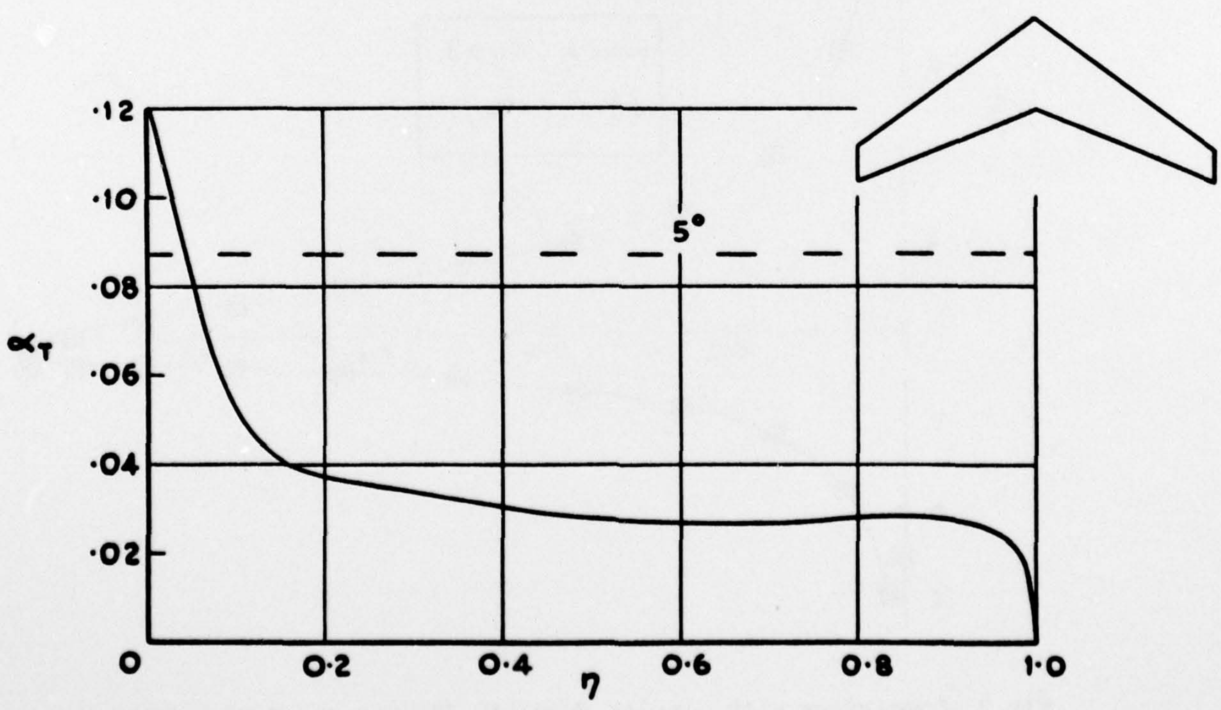


Fig. 4 Twist distribution on RAE wing 'B'

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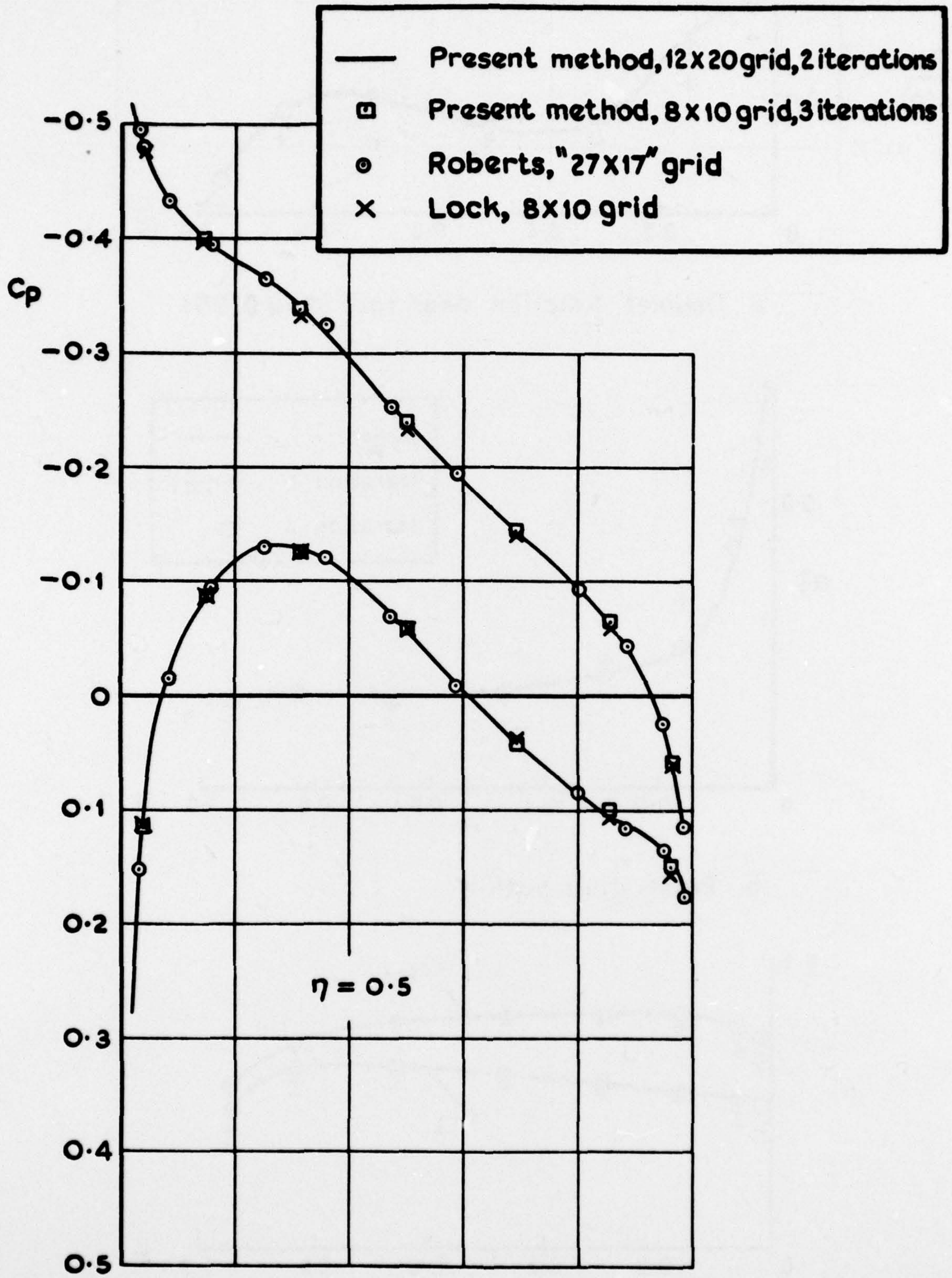
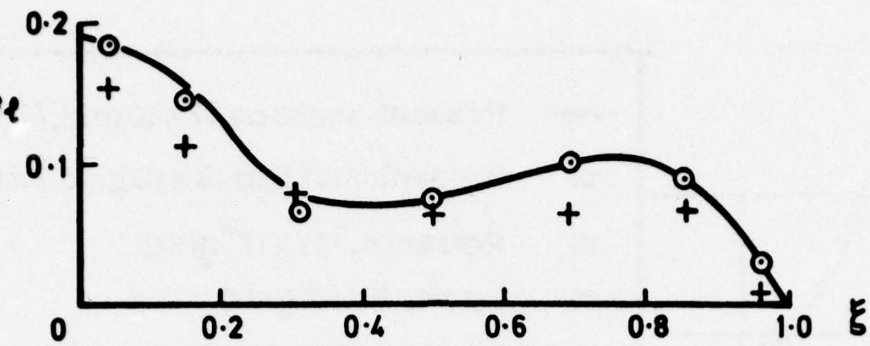
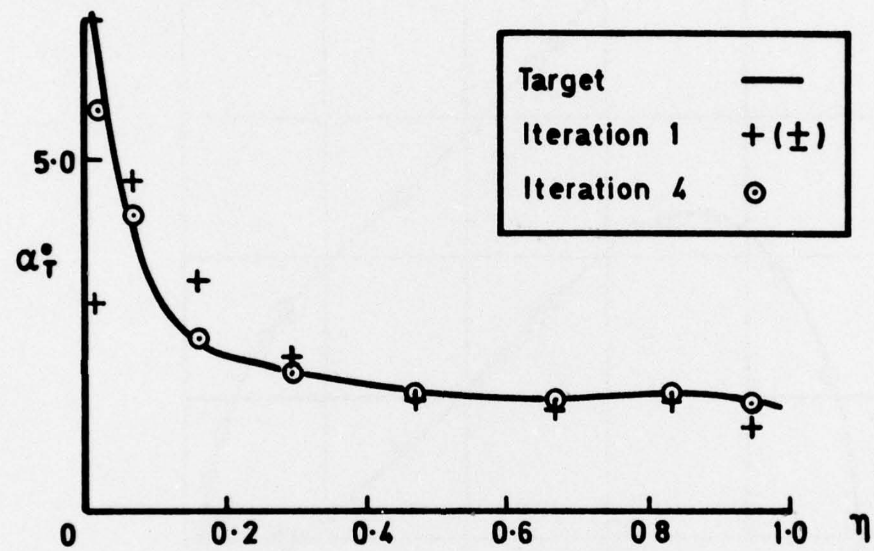


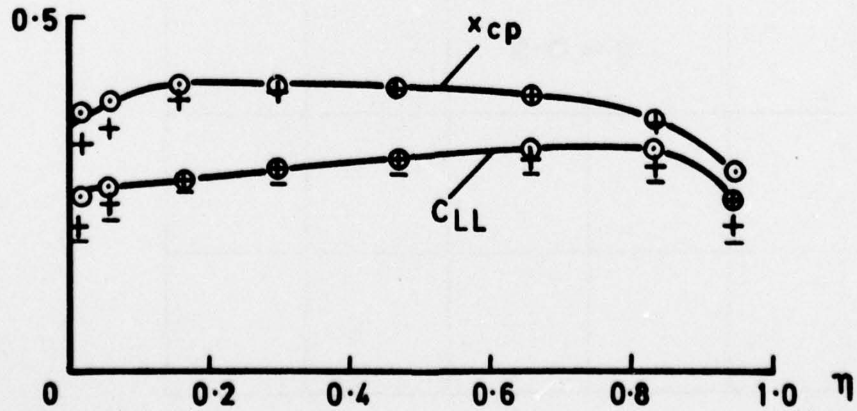
Fig. 5 Comparison of methods at mid-semi-span for RAE wing 'B' at zero incidence



a Doublet function near root (η=0·0155)



b Twist distribution



c Section lift and centre of pressure

Fig. 6a-c Iterates for RAE wing "B", M_∞=0·8. Problem 2

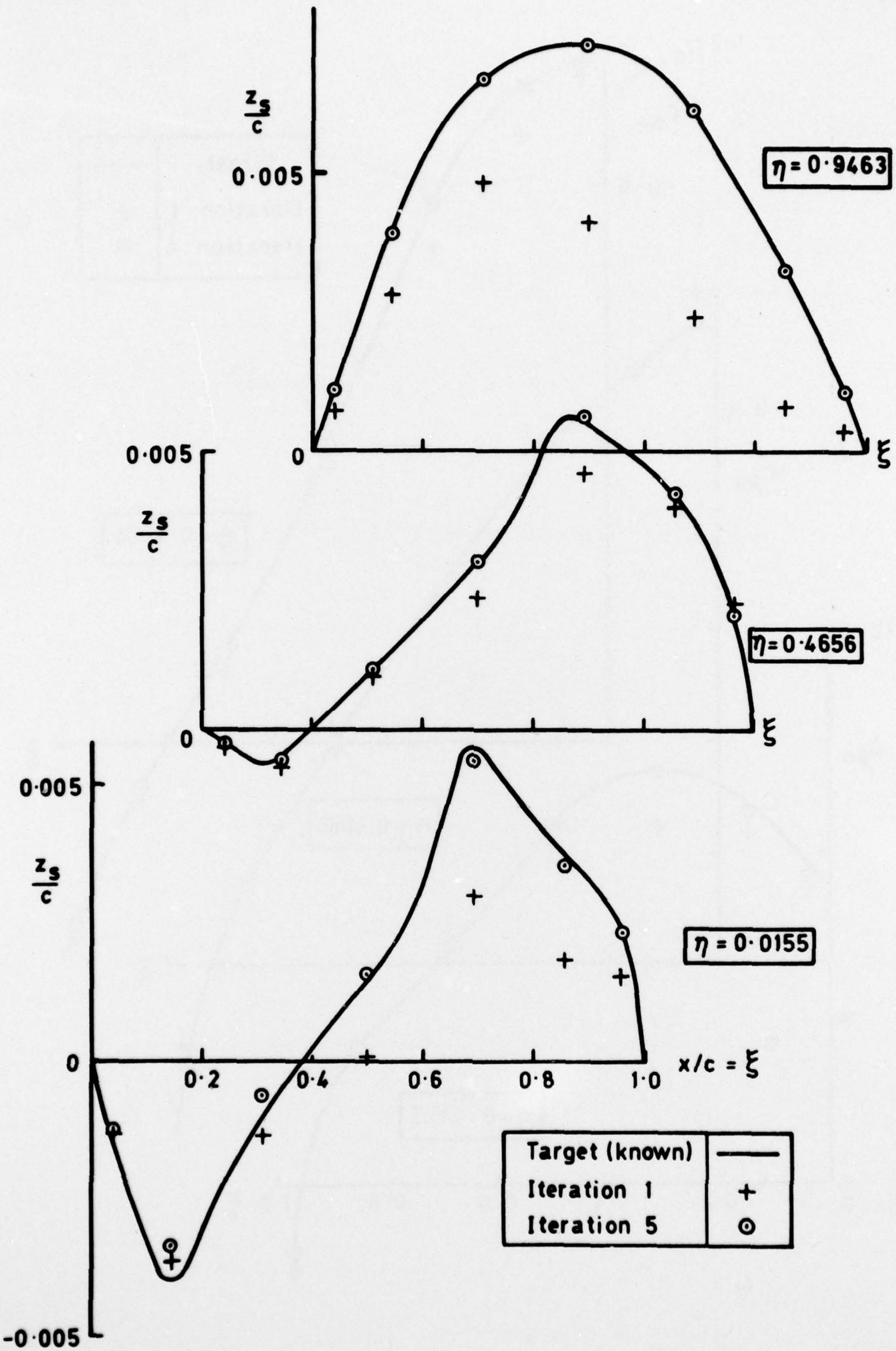


Fig. 7 Iterates for camber distribution on RAE wing "$\hat{B}$", $M_\infty = 0 \cdot 8$. Problem 4

Fig. 8

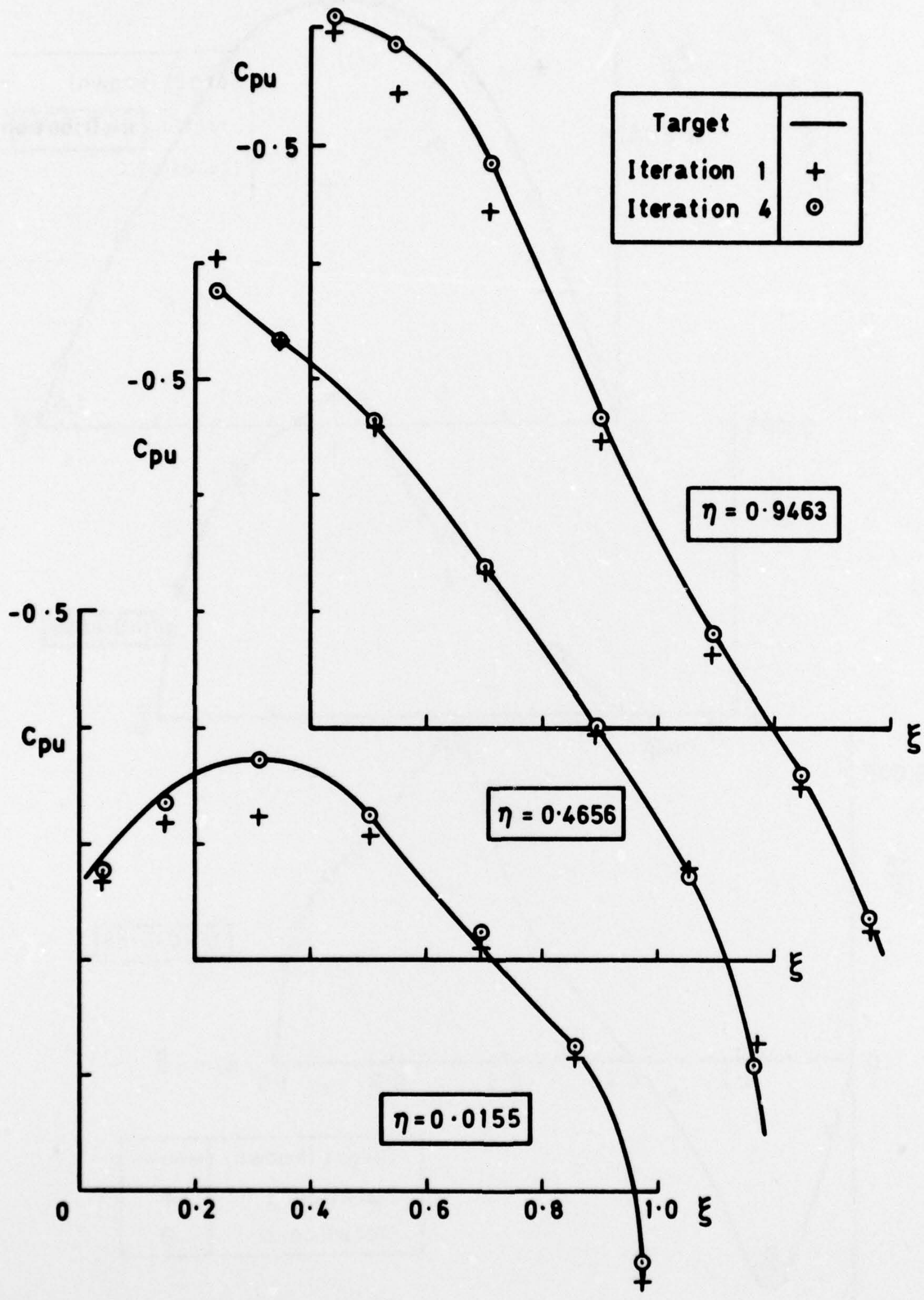


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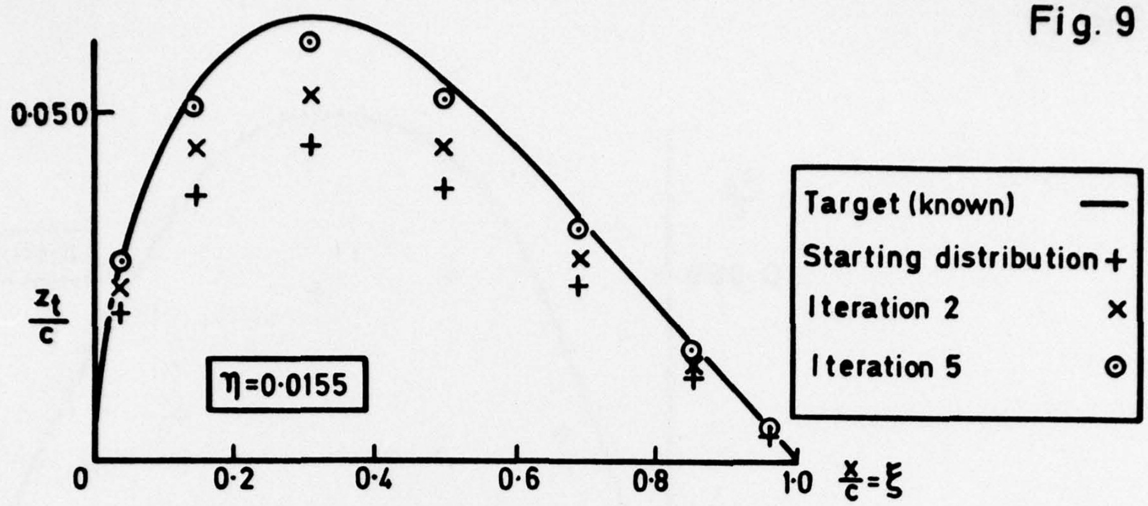
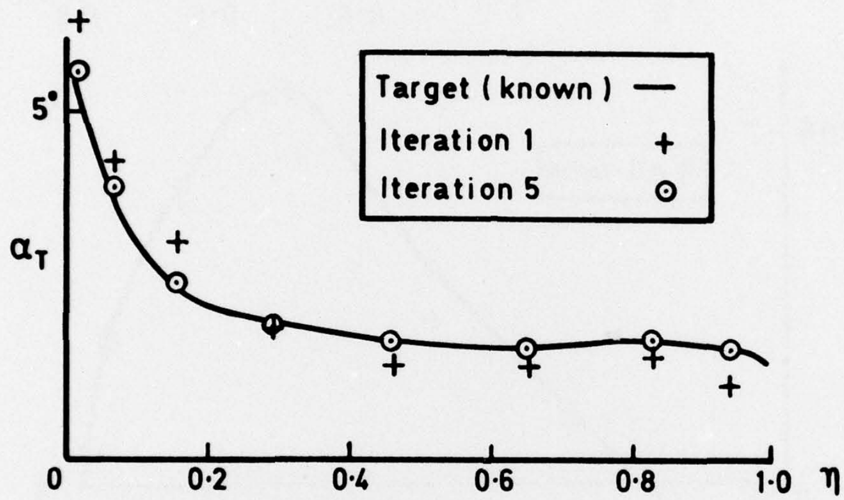
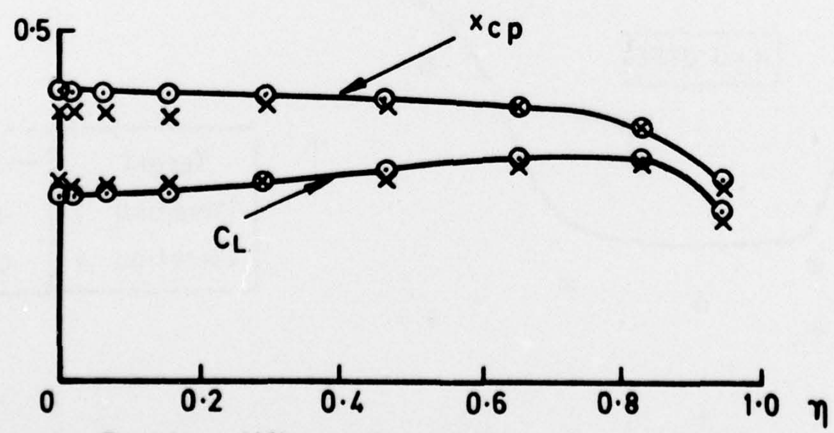
Fig. 8 Iterates for upper-surface pressure distribution on RAE wing "B", $M_\infty = 0 \cdot 8$. Problem 2



a Thickness distribution near root



b Twist distribution



c Section lift and centre of pressure

Fig. 9 Iterates for RAE wing "$\hat{B}$," M_∞ = 0·8. Problem 4

Fig. 10

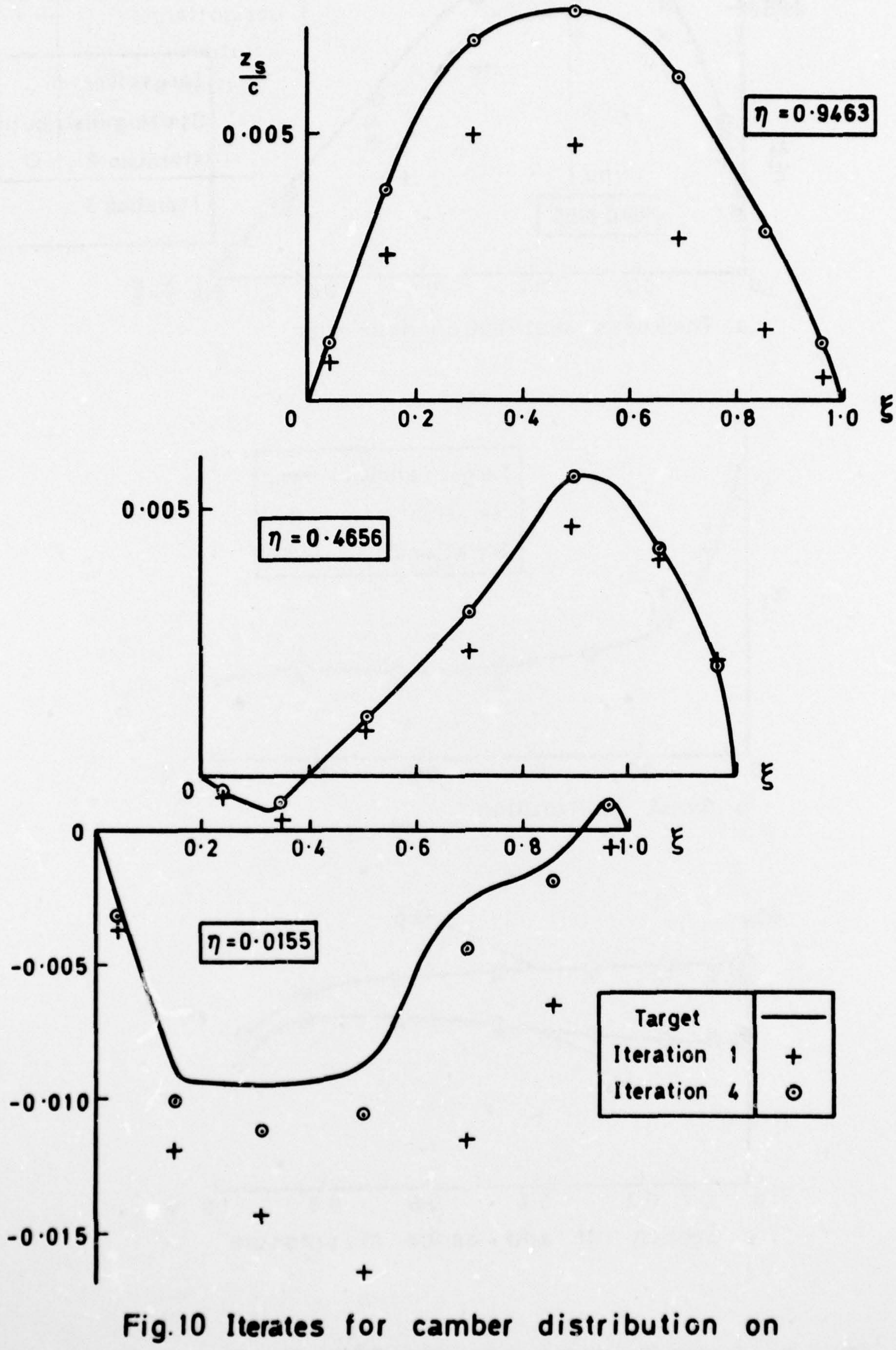


Fig. 10 Iterates for camber distribution on RAE wing "B", M_∞ = 0·8. Problem 2

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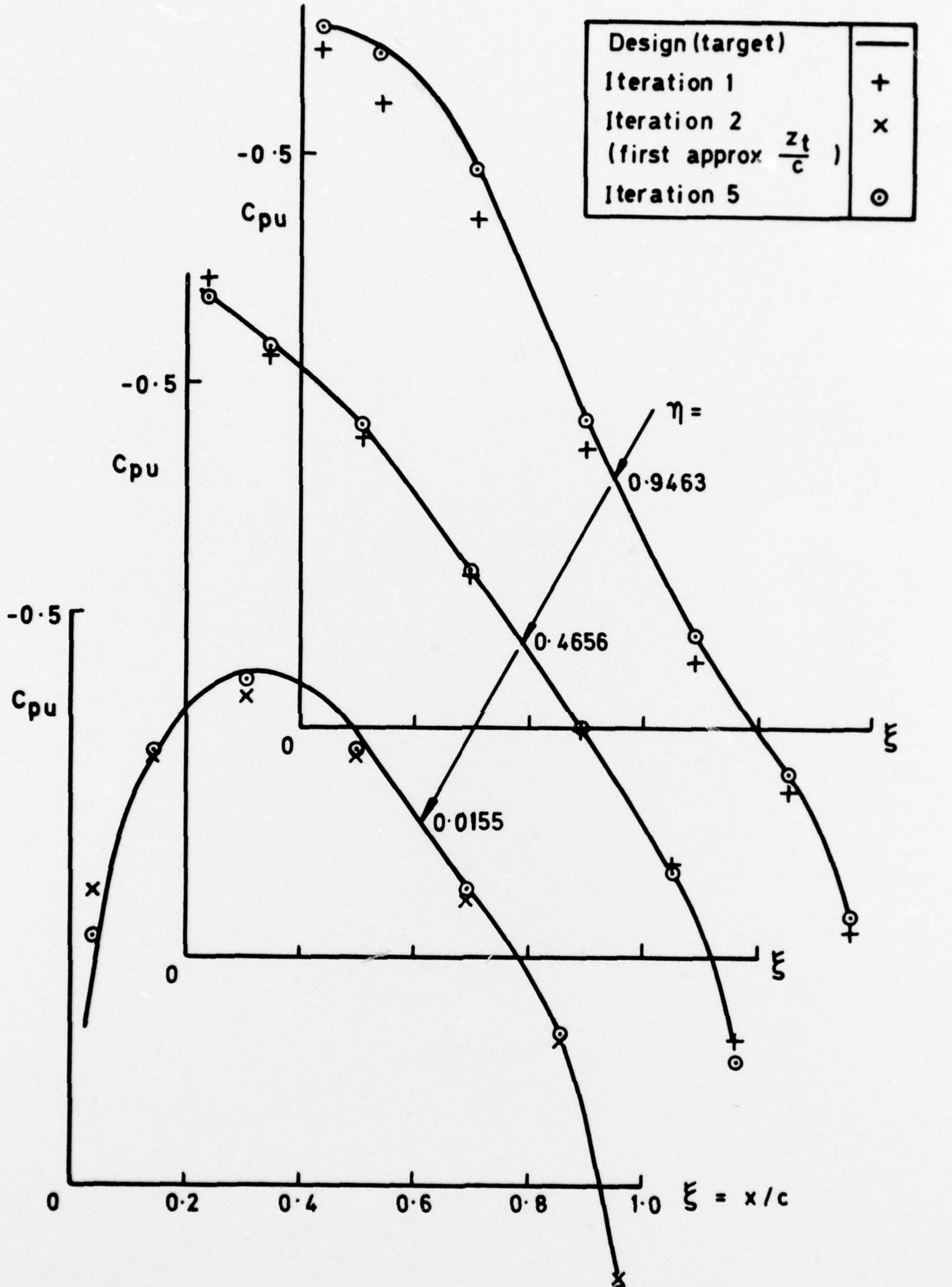

Fig. 11 Iterates for upper-surface pressure distribution on RAE wing "$\hat{B}$", M_∞ = 0·8. Problem 4

REPORT DOCUMENTATION PAGE

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17. Abstract

This is a written version of a lecture given by the author at Euromech Colloquium 75 which was held at Rhode, near Braunschweig in May 1976. The subject of the colloquium was 'The calculation of flow fields by panel methods' and this paper describes iterative calculations of the classic problems of steady inviscid flow around a thick cambered wing, providing improved accuracy in relation to the so-called RAE Standard Method.

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